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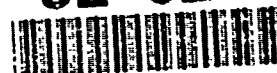
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A METHOD TO IMPROVE POST IMPACT COMPRESSIVE STRENGTH IN GRAPHITE/EPOXY COMPOSITE MATERIALS AFTER LOW TEMPERATURE IMPACT

A Trident Scholar Project Report

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A study was performed to investigate a method to improve the post impact compressive strength in graphite/epoxy composite materials due to low temperature impact. Aerospace quality 32 ply quasi-isotropic IM6/3501-6 test panels were utilized in the study. These panels were modified by adding FM300 interleafs, which have greater impact toughness, over the zero degree plies in four different locations in the symmetric layups. Ultrasonic C-scan inspections were used to characterize the quality of the panels, and provided a basis for comparison after the panels were impacted. The panels were impacted at -67 deg F by an instrumented impactor that produced load versus time and energy versus time curves. A compression after impact (CAI) test was performed to determine the post impact compressive strength of the panel. The results of the compression after impact test, post impact ultrasonic inspection, and optical microscopy were used to determine fracture mode and effects of interleafs.

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Abstract

A study was performed to investigate a method to improve the post impact compressive strength in graphite/epoxy composite materials due to low temperature impact. Aerospace quality 32 ply quasi-isotropic IM6/3501-6 test panels were utilized in the study. These panels were modified by adding FM300 interleafs, which have greater impact toughness, over the zero degree plies in four different locations in the symmetric layups. Ultrasonic C-scan inspections were used to characterize the quality of the panels, and provided a basis for comparison after the panels were impacted. The panels were impacted at -67 deg F by an instrumented impactor that produced load versus time and energy versus time curves. A compression after impact (CAI) test was performed to determine the post impact compressive strength of the panel. The results of the compression after impact test, post impact ultrasonic inspection, and optical microscopy were used to determine fracture mode and effects of interleafs.

The interleafed FM300 plies in each panel slightly decreased the post impact compressive strength of the material over the unmodified control panel. The interleafs did decrease the damage area compared to the unmodified panels, as characterized by C-scan inspections. The low temperature impact of unmodified IM6/3501-6 did not produce the predicted decrease in post impact compressive strength when compared to room temperature impact of identical specimens. The residual impact energy as a percentage of initial impact energy was approximately 85% for each of the four modified layups and the unmodified layup, indicating that the damage calculated by the energy versus time plot was the same for all five variations of the layup.

Acknowledgments

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Introduction

Current trends in high performance aircraft design require structural materials that have very high strength to weight and stiffness to weight ratios. Additionally, the material must be resistant to impact damage and corrosion. Many aerospace companies have recognized that these properties can be attained by composites, and have begun to use composite materials in areas ranging from aircraft control surfaces to fuselage panels. Military aircraft such as the F/A-18, C-5B, and the SH-60B all contain appreciable amounts of composite materials. In fact, the SH-60B helicopter contains over 17% by weight of composite materials in its construction.[1] The increased use of composites in the aviation industry enhances fuel economy and cargo capacity of many aircraft.

Background

Both natural and man made (synthetic) composite materials have existed for many centuries. Examples of natural composite materials include wood, bone, and teeth. One of the first examples of a primitive synthetic composite is straw reinforced clay bricks that the Israelites made in 800 B.C.[2] Straw was added to the strong brittle clay in order to help prevent cracking during drying and building. More recent historical examples include the use of fiberglass for radomes of aircraft during World War II.[3] Today composite materials

are used in many applications such as airplanes, boats, automobiles, bicycles, and sporting goods. Each of these applications requires a different type of composite material in order to fulfill successfully the design parameters.

In order to understand fully the advantages of composite materials and the benefits of this project, it is necessary to explain some terms relating to composite materials. As described by Schwartz[4], "A composite material is a materials system composed of a mixture or combination of two or more macroconstituents differing in form and/or material composition that are essentially insoluble in each other." In reinforced concrete the first constituent is the load bearing base material, cement, which is relatively weak under tension. The second constituent is the reinforcing material, such as steel mesh, which improves the tensile strength of the composite. There are five main types of composite materials, classified by the form that the reinforcing constituent takes. Figure 1 shows each class of composite material, with an example of each.

Individual constituents are combined to gain the benefits of each material, while offsetting negative properties of each. Some of the desirable properties are strength, ductility, and stiffness. Negative characteristics of each material may include a low melting point, brittleness, and low strength.
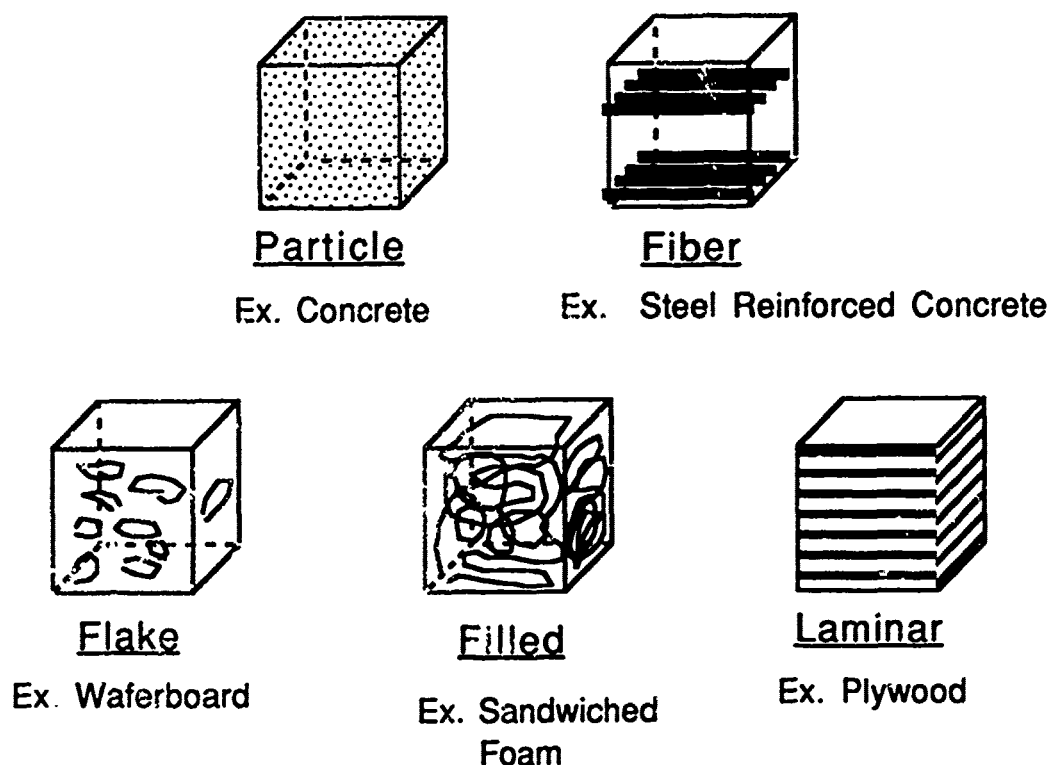
Figure 1 - Classes of Composite Materials

For all their benefits however, composite materials have a distinct disadvantage over metals in structural applications. The fatigue resistance and structural integrity cf composites are greatly decreased when they are damaged. It has been shown that a relatively low impact, such as a mechanic dropping a wrench on an aircraft wing, can reduce its compressive strength by as much as 50%. [5] In order to enhance the safety of composite construction in impact prone areas, it is necessary to protect the material from the damaging effects of impacts.

The impact properties of composites are affected by the composite matrix, fiber type, and fiber orientation. The matrix is the material in the composite that surrounds the fibers. Polymer matrices fall into two broad categories: thermosets and thermoplastics. Thermosets, such as plexiglass, are hard brittle polymers with a high tensile strength. They are particularly suited for high temperature applications because they do not soften and melt. Thermoplastics, such as polyethylene refuse bags, are much more impact resistant and tough. They do not have the tensile strength of the thermosets and they will deform and melt at relative low temperatures.

Most aerospace composite materials consist of laminates which have fiber reinforced polymer matrices. Methods of reinforcement include adding short chopped fibers, long discontinuous fibers, or long continuous fibers. Since the strength of the composite plies are greatest parallel to the fiber direction, many composite layups will vary the orientation of the unidirectional fibers to get quasi-isotropic properties. Quasi-isotropic means that the mechanical properties of the composite panel do not vary with direction in the plane of the ply.

An example stacking sequence for a 16 ply panel is [+45, 0, -45, 90]$_{2S}$. In this sequence, the first ply is aligned with its unidirectional fibers along a 45 degree direction with respect to a reference direction. The second ply is aligned along a 0 degree direction, the third at -45 degrees, and the fourth at 90 degrees. This

sequence is repeated two times, as indicated by the subscript 2. The sequence then reverses and continues to the 16th ply. This creates a symmetric layup, designated by the subscript "S" after the bracketed sequence, about the midplane of the panel. Figure 2 shows an example of this technique on a 16 ply panel, to achieve quasi-isotropic properties.

Examples of reinforcing fibers include carbon, glass, and aramid fibers. Kevlar is the trade name for an aramid reinforcing fiber widely used in bulletproof vests and tires. The oldest fiber reinforced composites employed glass fibers, and have been in use since the early 1940's.[3] Higher strength composites are possible through the use of high strength fibers such as Kevlar and carbon. Glass fibers, although still attractive because of their low cost, have a lower strength to weight ratio than either carbon or Kevlar fibers.

```
        45
1st sequence    0
        -45
        90
        45
        0
. 2nd sequence  -45
        90
        90
        -45
        0
        45
        90
        -45
        0
        45
```

Two sequences symmetric about midplane.

[+45, 0, -45, 90] $_{2s}$

16 ply layup
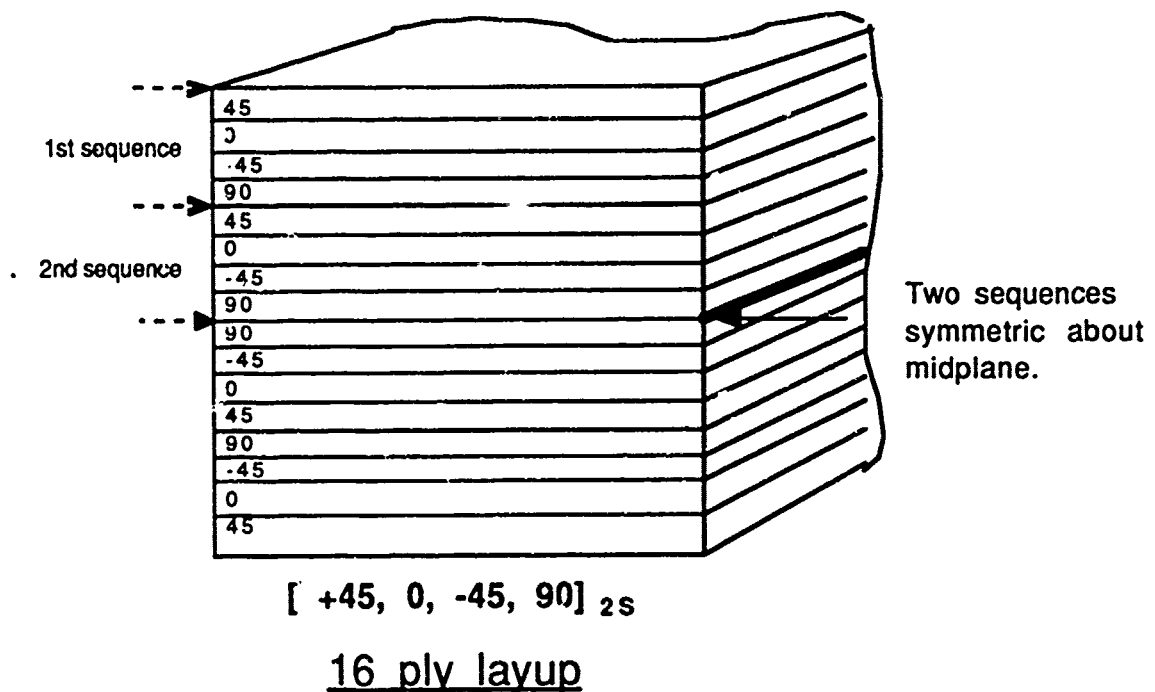
Figure 2 - An Example Stacking Pattern to Achieve a Quasi-Isotropic Panel

Composite materials are constructed by two main methods: prepregs and wet layups. A prepreg is a flexible sheet containing one uncured ply of the layup, consisting of the fiber reinforcement surrounded by the matrix material. Backing paper is placed between prepreg plies during transportation and storage to keep them from sticking to each other. During construction, the backing paper is peeled off the prepreg after it is placed on the layup stack with the proper fiber orientation. The completed stack of prepreg sheets, that create a composite panel, are placed in an autoclave to cure the epoxy matrix under heat and pressure. The advantages of this

method are the relative ease and speed of panel layup, and the smaller probability of having porosity in the panel after curing.

The second method, called a wet layup, is much more time consuming. The dry fiber reinforcement is cut and placed with the proper orientation on the stack. The resin matrix is then poured on top. The resin is forced into the dry reinforcement and the excess is removed. This method produces a panel that usually contains more porosity than the prepreg method. The porosity can lead to low impact toughness.

The quest for highly impact resistant composite materials has been addressed in several interesting ways. One method is to develop matrix systems that are impact resistant. [6-8] This method has been successful, but it has been shown that a tenfold increase in resin toughness only increases the composite toughness by a factor of two. [9] Also, while tough resins reduce the extent of impact damage, they frequently cause a reduction in composite strength because toughness is obtained at a sacrifice of stiffness. [10] One significant benefit of tougher resins is that their use allows for a reduction in manufacturing costs through the use of thicker plies. Tests have shown only a slight reduction in strength is observed when ply thickness is increased from 0.127 mm to 0.508 mm. These thick plies sharply reduce the layup time for a thick structural part. [11] Another

method to increase the impact toughness is to increase the tensile strength of the reinforcing fibers. It has been shown that for any impact energy, stronger fibers show decreased damage and increased post impact compressive strength. [12] The most promising method is to add a layer or interleaf of tough impact resistant material between the plies of the composite material as it is being laid up. The interleaf acts like a shock absorber and helps dissipate the impact energy, while protecting the plies of the composite panel below it. The advantages of this method are the ease with which it can be incorporated into existing fabrication facilities, and the possibility for tremendous increases in the post impact compressive strength of the composite system. Tests have already been done to show that interleafing tough plies into a composite layup can double its post impact compressive strength compared to an unmodified panel. [13] All of these experiments have been carried out at room temperature, where the properties of the interleaf are predictable. When composites are subjected to low temperatures, they tend to get increasingly brittle. This is a definite problem for military aircraft operating in arctic environments or the low temperatures of high altitude flight. It is not known if the addition of an interleaf will control impact damage and/or enhance post impact compressive strength at low temperatures.

Objectives

The impact resistance and damage tolerance of thermoset polymer matrix composites require improvement, especially at arctic temperatures. Interleafing of a tough ply could provide improved impact resistance and damage tolerance in a multi laminae composite. The objective of this project was to investigate the use of an interleaf on composite impact toughness and post impact compressive strength at low temperature. The specific objectives, therefore, were: (1) determine the effect of the addition of a tough interleaf on the toughness of the composite material, (2) determine the optimum placement for the interleafs to maximize the post impact compressive strength, (3) determine the effect of the introduction of tough interleafs on the impact damage area, and (4) investigate the failure mechanisms in the impact area. In order to provide the low temperature environment for the project, an environmental chamber had to be designed and built.

The results from this research project in composite toughening could have widespread applications in both commercial and military interests. The increased resistance to impact damage would allow greater use of composite materials in aircraft, automobiles, and ships, with levels of safety and reliability that were previously unattainable.

Experimental Details

Materials

In order to increase the impact resistance of an aerospace composite material, this project originally sought to combine the strength of a thermosetting matrix with the impact toughness of a thermoplastic interleaf. Polyetheretherketone (PEEK), a tough thermoplastic, was to be interleafed into an existing aerospace thermoset matrix composite system. An inquiry of industrial sources was made as to the possibility of bonding thermoplastic PEEK to the thermoset matrix composite. A method to bond the thermoset matrix with the thermoplastic interleaf could not be found. To overcome this obstacle in the program an impact resistant thermosetting material was used instead, to avoid the bonding problems between two different polymer types.

In this study an existing composite system, Hercules IM6/3501-6 (fiber / matrix system), was modified by the addition of a toughened interleaf, FM300, at various locations. FM300 is a modified high strain tough epoxy, developed to reduce shear stress concentrations in composite structures.[14] The location of the interleafs was varied to determine the optimum placement which maximized the post impact compressive strength and minimized the impact damage area.

Five 32 ply quasi-isotropic panels, constructed of IM6/3501-6, were obtained from Hercules Aerospace. IM6/3501-6 graphite prepreg tape is an amine cured epoxy resin, reinforced with unidirectional graphite fibers. The reinforcements are Hercules continuous type IM6 graphite fibers, surface treated to increase the composite shear and transverse tensile strength. Hercules 3501-6 was developed to operate in a temperature environment as high as 350 deg F.[15] All five plates were constructed with a stacking sequence of [+ 45, 0, - 45, 90]$_{4S}$. The mechanical properties of the unmodified IM6/3501-6 composite are given in Figure 3.[15]

Magnamite ® Graphite Prepreg Tape
IM6/3501-6

0° Tensile Strength	370,000 psi
0° Compression Strength	229,000 psi
0° Tensile Modulus	24.0×10^6 psi
CAI* strength after 1500 in-lb/in impact	22,000 psi
Fiber Volume	62 %
Cured Ply Thickness	5.5 mils
Fiber Areal Weight	145 g/m^2

* Compression After Impact (CAI)

Figure 3 - Mechanical Properties of IM6/3501-6 Prepreg Tape

One of the panels was unmodified, and served as the control panel. The other four panels were modified by adding an interleaf of FM300, a tough thermoset, over the first zero degree plies in the first modified panel. The second modified panel contained FM300 interleafs over the second zero degree plies. This was repeated for the third and fourth zero degree plies in the last two modified panels. Since uniaxial force is applied along the zero degree fiber direction, these locations were chosen because the zero degree plies contribute the most to the compressive strength in the compression after impact (CAI) test. It was hoped that the placement of the FM300 over these plies would help protect them and increase their structural integrity. The location was varied in each of the four panels in order to determine the optimum placement of the interleafs. The location of the interleafs in the modified panels is shown in Figure 4. Plate 1, which is not shown, is the unmodified control plate. Plates 2-5 are the modified panels with the interleafs located as indicated in Figure 4.
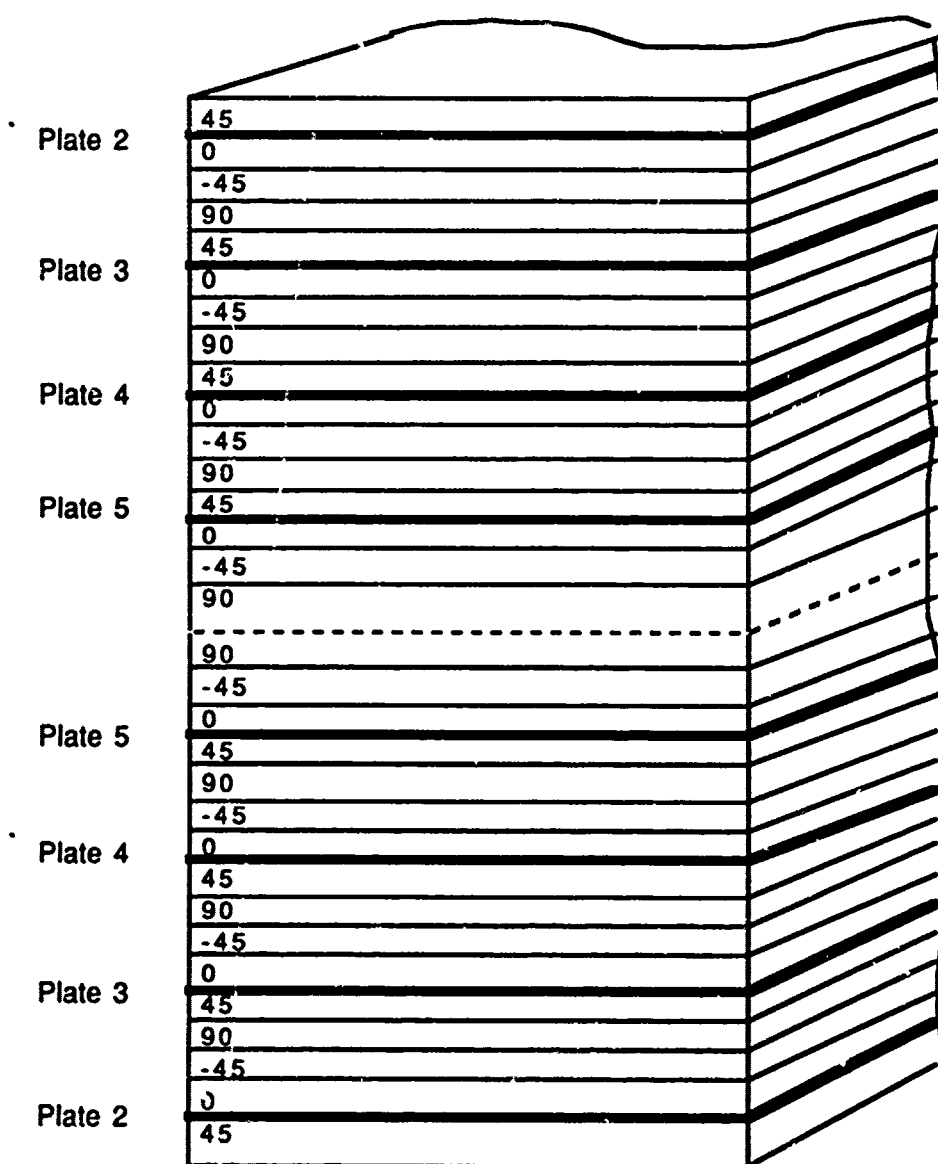
Figure 4 - Placement of FM300 Interleafs in Modified Plates

The panels were constructed from 32 layers of prepreg tape, with approximately 62% fiber volume. After the panels were laid up, they were cured in an autoclave, as shown in Figure 5, to achieve their maximum strength. An autoclave uses high temperature and pressure to cure the epoxy matrix. The uncured panel is covered with release plies and placed between two platens. The platens are covered with a vacuum bag and the edges are sealed to maintain the vacuum when it is applied. When a vacuum is applied to the vacuum bag, it collapses to apply pressure, via the platens. on the panel. The vacuum also helps remove any air that would be trapped, causing voids in the panel. The air inlet introduces high pressure air on the outside of the vacuum bag to increase the pressure on the panel. Heat is added to cure the epoxy. A schematic of the cure cycle for the autoclave is indicated in Figure 6.

Typical Autoclave
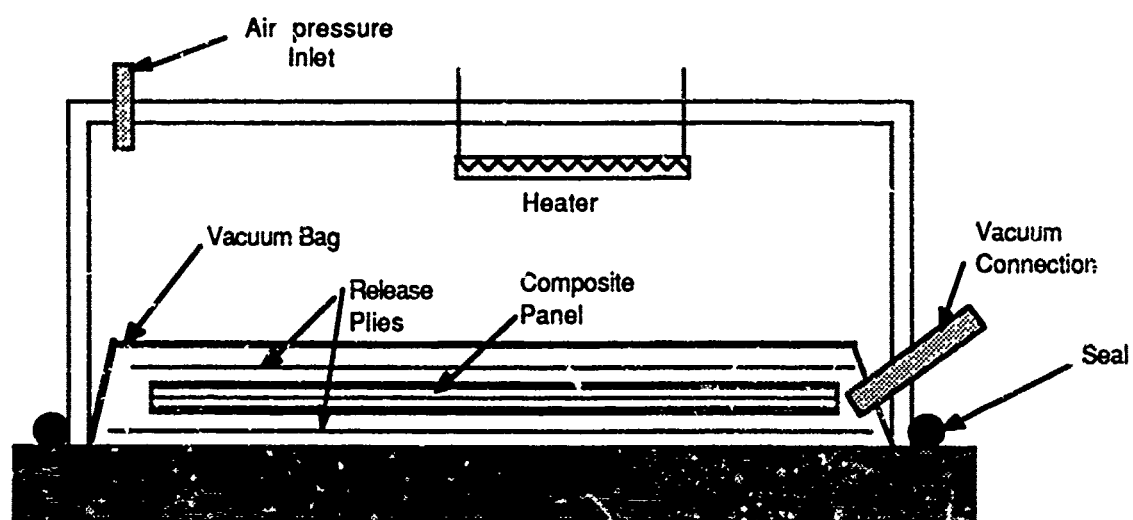
Figure 5 - Schematic of an Autoclave For Composite Panel Production

1. Panel is enclosed in a vacuum bag and placed in the autoclave.
2. Header vacuum of 20 inches Hg is applied to vacuum bag.
3. Autoclave is pressurized to 90 psig.
4. Panel is heated at 5 deg F per minute to 355 F.
5. Vacuum bag is vented when pressure reaches 20 psig.
6. Panel is held at 355 F for 130 minutes.
7. Panel is cooled at a rate of 5 deg F per minute to 140 F.
8. When panel cools below 140 F, autoclave pressure can be vented and panel can be removed.

Figure 6 - Typical Cure Cycle For Composite Panel Production

Pre - impact Ultrasonic Inspection

The test materials were received in the form of 14 X 15 inch panels from Hercules Aerospace. Before each panel was cut into six 4 X 6 inch impact specimens, it was necessary to determine the absence of defects within the composite layup. In order to determine if defects were present, ultrasonic nondestructive evaluation was employed. Ultrasonic inspection of the material is accomplished by using a high frequency sound wave from a piezoelectric transducer. The

frequencies utilized for ultrasonic inspection usually range between 0.5 and 10 Mhz. A 5 Mhz, 0.75 inch diameter focused transducer was used for these inspections. The sound wave is transferred from the transducer to the specimen through a couplant material such as water. This couplant is necessary because the characteristic impedance of the graphite/epoxy panel is approximately 100,000 times that of air, which would cause an almost total reflection of the incident wave as it struck the top surface of the panel.[16-17]

Reflections are created as the incident wave strikes the front and back surfaces of the specimen, as well as any defects within the specimen. The reflections are caused by the difference in characteristic impedance between the uniform specimen and the defect. The reflections created at the front and back walls are due to the impedance difference between the specimen material and the water.

There are three standard ways that the series of pulses and echoes from an ultrasonic inspection can be represented. These are called either A, B, or C-scan displays. The specifics of each method are described in the following text.

Side View
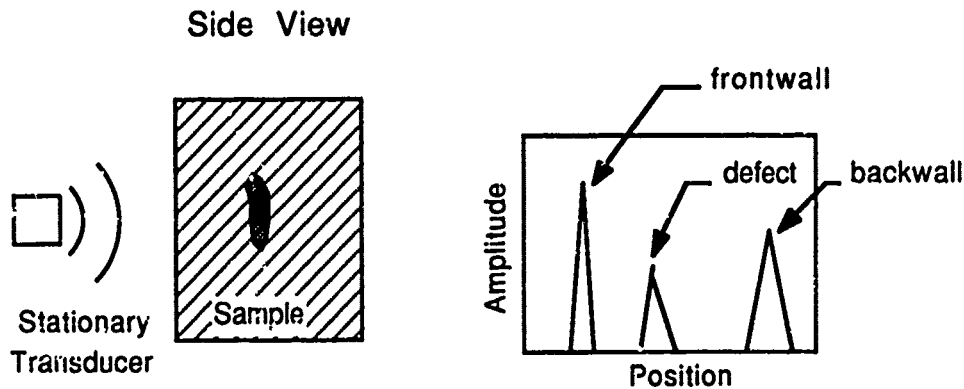
Figure 7 - A-scan Representation of Ultrasonic Data

As Figure 7 shows, in the A-scan representation, the reflections are shown as a series of peaks from a baseline noise level. The amplitude of the peak roughly corresponds with the defect size, in that larger defects will have have a larger amplitude than smaller ones. The location of an echo with respect to the large front and back wall echoes gives its relative position within the specimen, thus a defect echo appearing halfway between the front and back wall echoes would be located near the midplane of the panel. One major drawback for the A-scan is that it only represents the portion of the specimen directly underneath the current position of the transducer.
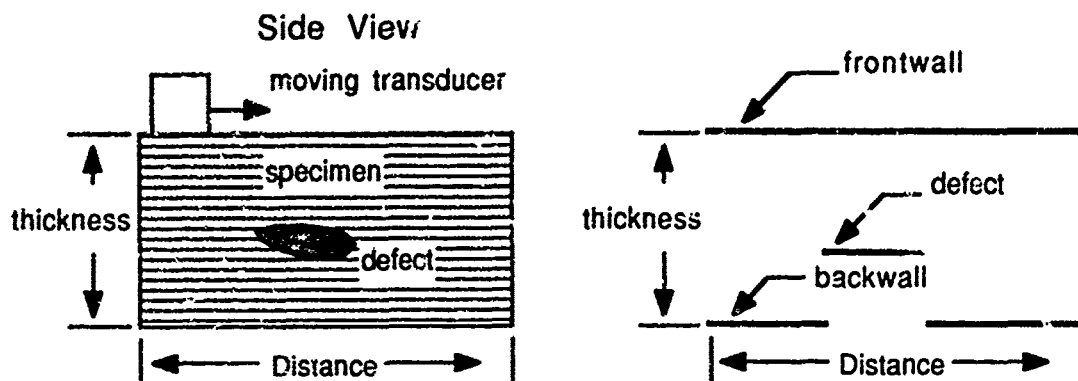
Figure 8 - B-scan Representation of Ultrasonic Data

Figure 8 shows the B-scan representation for ultrasonic data. This method uses a transducer moving in a line at a constant rate across the specimen. As the wall and defect echoes are received, they are shown as a series of lines. The pattern of the lines indicate the relative length and depth of defects. This representation can be compared to what the specimen would look like if it were cut along the path of transducer movement and visually inspected. The drawback of the B-scan is that it only gives the relative length and location of defects within one thin slice of the specimen.
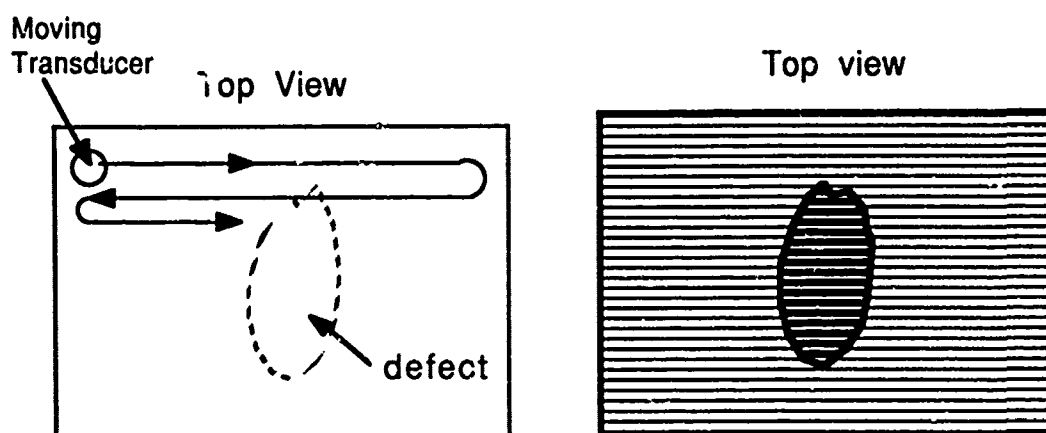
Figure 9 - C-scan Representation of Ultrasonic Data

Figure 9 shows a C-scan representation of the ultrasonic data. This method integrates a series of transducer passes, giving a plan view of defect location and size within the specimen. C-scan data can be collected in two different ways. The first method uses only one transducer, and is termed the pulse-echo method. The single transducer produces an ultrasonic sound pulse then pauses to collect the echoes. The pulse repetition frequency of the transducer is adjusted to ensure that the transducer does not pulse again until the back wall specimen is received. An added benefit for this method is that since the sound velocity in the specimen is known, as well as the time intervals in which the defect reflections are received, the defect depth can be calculated by the signal processor. This feature allows a "time of flight" representation which is a plan view as before, but defects are shaded differently according to their depth within the

specimen.

The second collection method, called "through transmission", uses a transducer on each side of the specimen. One transducer constantly transmits while the other continually receives the strength of the sound pulse on the other side of the specimen. The strength of the signal passing through the specimen is decreased by defects reflecting and attenuating the wave energy. A plot of the amplitude of this signal allows comparison between solid sections and the defective sections in a panel. The solid sections are characterized by a large percentage of ultrasonic transmission, whereas the defective sections are characterized by minimal ultrasonic transmission. This method is somewhat more complicated because two transducers must be moved in unison back and forth across the specimen. The slight advantage offered by this method is that it allows the examination of thicker specimens where attenuation would be a problem for a pulse echo system.

The C-scan system used to inspect all panels in this investigation used the pulse echo method with a 5 Mhz transducer. The panel was submerged in a water tub and held off the bottom with aluminum rods at the edges. The transducer transversed the entire panel through the use of an X-Y servo motor system. The C-scan software allowed simultaneous pulse-echo and time of flight representations. Additionally, through the use of a signal gate on the backwall echo, a

through transmission representation was also recorded.

After the panel was impacted, another C-scan was performed to investigate the effects of the interleafs on damage area. The following figures show representative pre and post-impact C-scan images of the panels. In Figure 10A, the pulse echo representation, notice that the C-scan detected some of the fiber orientation which appears as the white patches along 45 degree directions across the panel. Additionally, the slight decrease of 0.001 inch of thickness near the upper edge is also detected In Figure 10B, the through transmission representation, the strength of the signal passing through the undamaged panel is constant. On this scale, the dark color indicates a large percentage of the signal is passing through the panel, indicating uniformity and the absence of major defects within the panel. Example C-scans of damaged areas are shown in Figure 11. Figure 11A is the pulse echo picture, which shows damage area within the specimen. Figure 11B is the through transmission picture, which shows that the damage in the center completely attenuated the signal, Additionally, the delaminations along the 45 degree ply on the reverse side of the panel are detected as shown.
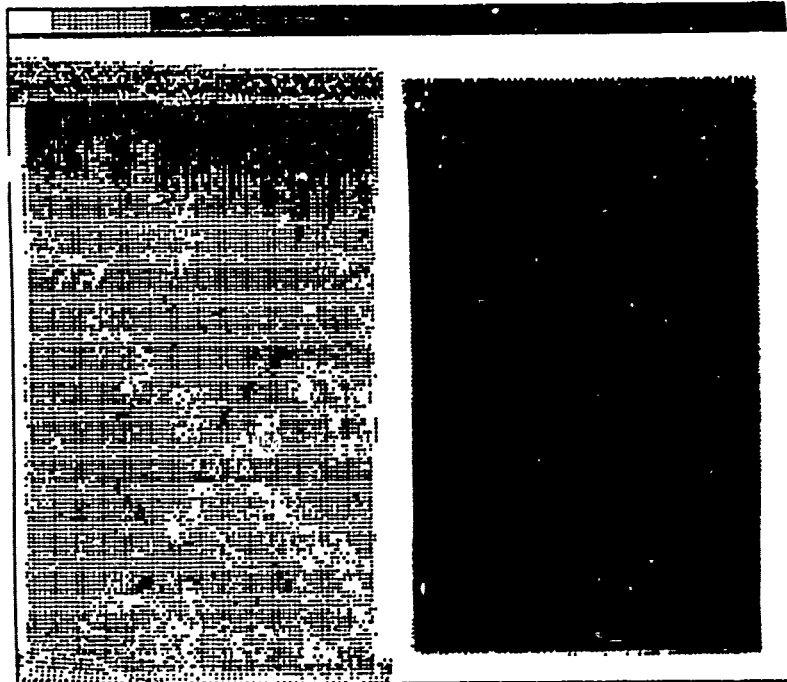
Figure 10A   Figure 10B

Figure 10 - Ultrasonic Plots of Pre-Impact Specimens
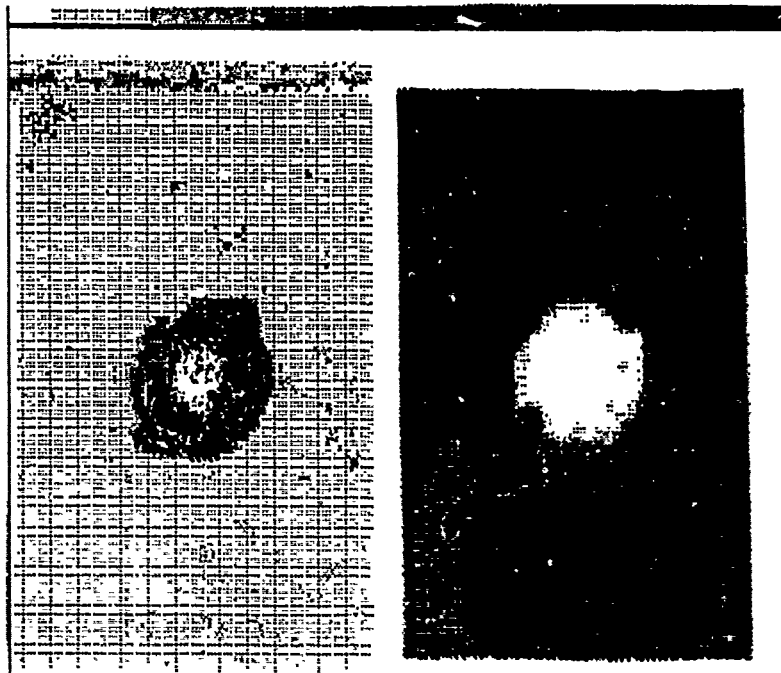
Figure 11A   Figure 11B

Figure 11 - Ultrasonic Plots of Post-Impact Specimens

Impact Testing

In order to characterize the impact toughness of the panels, an instrumented falling dart impact test was chosen. This test utilizes an spherical tipped dart containing a load cell to measure the impact force from the panel on the dart as a function of time. This force-time curve is integrated by computer to produce an absorbed energy versus time curve. Figure 12 shows the impactor. The mathematical analysis for the impact response is as follows: [18]

IMPACTOR

The output of the load cell, P(t), which is transmitted to the computer.

P(t)
The force of the panel on the impactor.
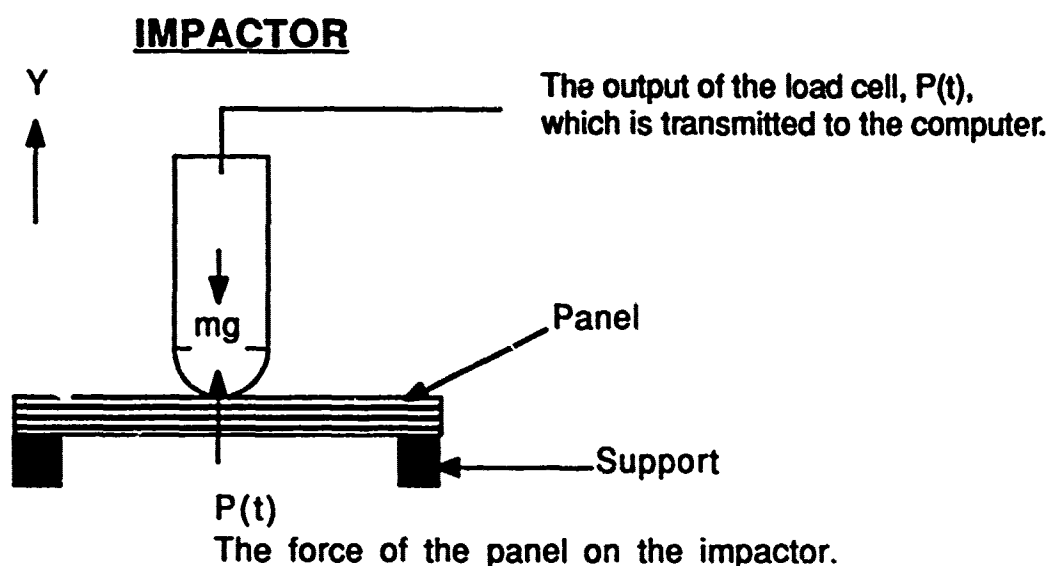
Figure 12 - Forces Acting on the Instrumented Impactor.

The total force on the impactor at time t is

$$f(t) = mg - P(t) \tag{1}$$

where mg is the force due to gravity and P(t) is the force due to panel impact. The acceleration, a(t), is found from

$$a(t) = f(t) / \text{impactor mass} \tag{2}$$

substituting f(t) from (1) into (2), the following is obtained

$$a(t) = g - P(t) / m \tag{3}$$

Equation (3) is integrated once to obtain the velocity, v(t)

$$v(t) = gt - (1/m) \int P(t) \, dt \tag{4}$$

Equation (4) is integrated again to obtain position, y(t).

$$y(t) = 1/2 \, gt^2 - (1/m) \iint P(t) \, dt \tag{5}$$

Conservation of energy in the impactor and panel during the test allows

$$E(t) = KE(t) + PE(t) + \text{Absorbed Energy (t)} = \text{Constant} \tag{6}$$

where E(t) is the total energy, which equals the initial kinetic energy of the impactor upon impact, KE(0). KE(t) and PE(t) are the kinetic and potential energies of the impactor, respectively. Equation (6) leads to

$$\text{Absorbed Energy (t)} = KE(0) - KE(t) - PE(t) \tag{7}$$

where

$$KE(t) = (1/2) \, mv^2(t) \tag{8}$$

since the impact creates a depression in the panel, decreasing the potential energy of the impact area,

$$PE(t) = - mg \, y(t) \tag{9}$$

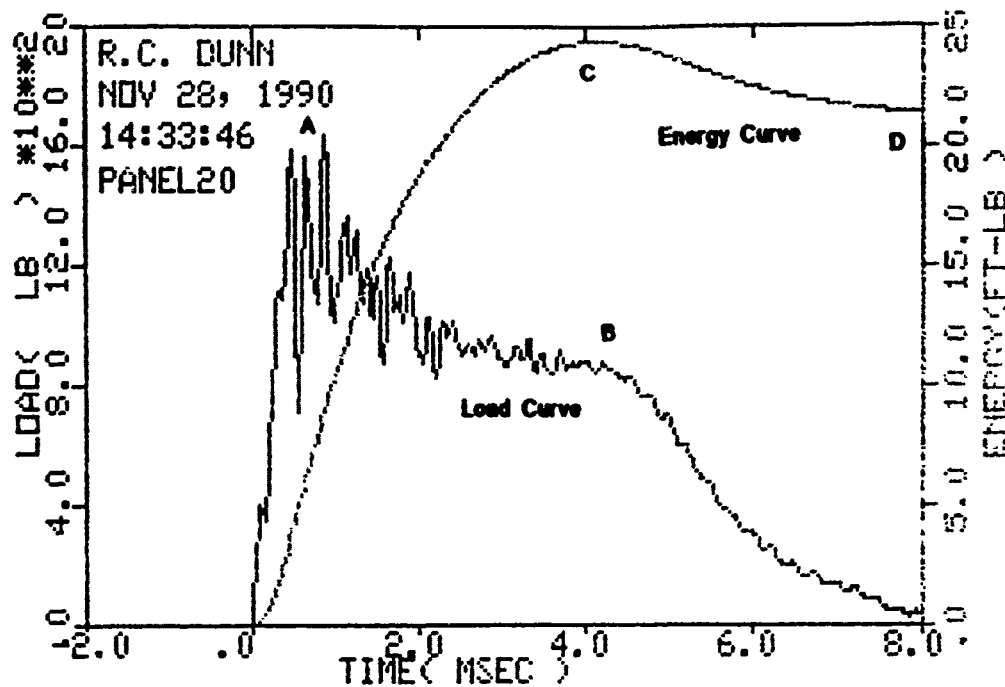therefore, by substituting (8) and (9) into (7), the following is obtained

$$\text{Absorbed Energy (t)} = (m/2)\ (V^2_{impact} - V^2(t)) + mg\ y(t) \qquad (10)$$

Equation (10) shows that the energy absorbed by the specimen is equal to the sum of two energy components. The first is the change in the kinetic energy of the impactor between the instant of impact and the instant the impactor leaves the panel surface during rebounding. The second component is the change in potential energy of the impact area, which is measured downward from the pre-impact panel surface. This test can be compared to dropping a weight on a spring and compressing it beyond its elastic limit. The weight will not rebound off the spring with its same initial kinetic energy. The permanent deformation in the spring is analogous to the energy absorbed as damage in the composite panel.

The signal from the load cell, P(t), is recorded by the computer in equal time intervals. The computer uses the trapezoidal rule to perform the integrations to calculate v(t) and y(t).[18] The physical meaning of the the area under the force-time curve is the variation in momentum (m dV) of the striker due to impact.[19] A typical force-time curve and its associated energy-time curve is shown in Figure 13. Significant points on both curves are labeled and described beneath the figure.

R.C. DUNN
NOV 28, 1990
14:33:46
PANEL20

Energy Curve

Load Curve

LOAD(LB) *10**2

ENERGY(FT-LB)

TIME(MSEC)

PointA: Maximum load

Points A to B: Fiber breakage, matrix cracking, and debonding

Point C: Maximum impact energy

Points C to D: Energy returned to impactor (Rebound)

Point D: Absorbed impact energy (Damage)

Figure 13 - Force vs. Time and Energy vs. Time Impact Curves

The impact testing of the panels was carried out following the guidelines set by the Suppliers of Advanced Composite Materials Association, (SACMA), recommended method SRM 2-88. [20] First, the impact energy per unit thickness must be se*. For this study 1500 in.-lbs/in. was selected. To determine this, the panel thickness is required as per the following equation.

in.-lbs/in. thickness = (Drop weight X Height) / Thickness.   (11)

The thickness, needed to calculate the required impact energy of 1500 in-lbs/in, was measured by an average of four measurements taken with a ball nose micrometer around the impact area.

Each of the five different plates used in this study were cut into six 4 X 6 inch impact specimens. All six specimens from each plate were impact tested on a Dynatup model 8200 drop weight impact tower with a 0.625 in diameter spherical tip impactor as shown in Figure 14. The weight of the impactor system was 10.86 lbs. The average specimen thickness was 0.188 in, which translated into average drop heights of 25.9 in. The drop height for the impactor was adjusted slightly on each test to ensure the desired 1500 in.-lbs/in impact energy was obtained. The impactor is attached to a frame that slides on the vertical rails shown in the figure. This figure also shows how the specimen was constrained at each corner over a 3 X 5 inch cutout by a Boeing restraint fixture.

Instrumented Tup

Restraint Clamps (4)

4 X 6 in. Impact Specimen

Boeing Restraint Fixture
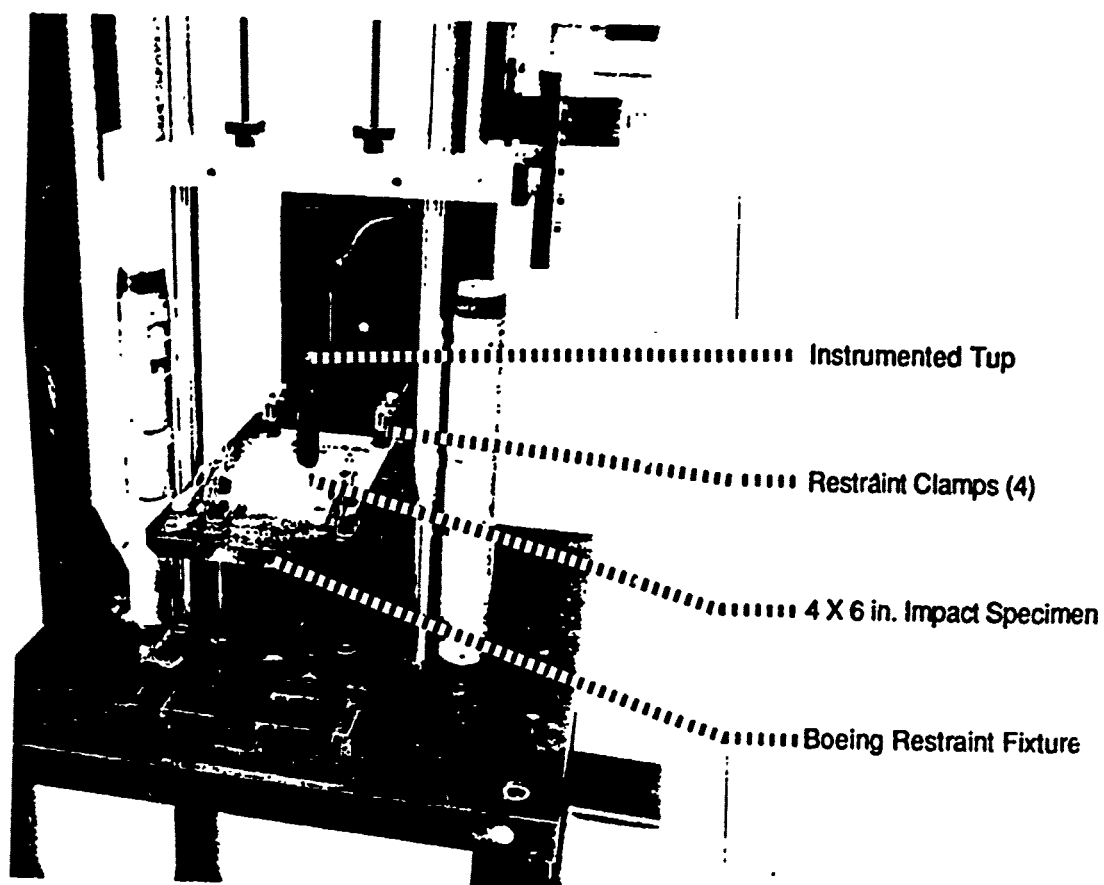
Figure 14 - Impact Tower and Instrumented Impactor

Environmental Chamber

In order to achieve a military specification temperature of -67 deg F, the base of the impact tower and the Boeing test fixture were enclosed in the environmental chamber as shown in Figure 15. This chamber was constructed from high density polystyrene foam panels, cut and fitted around the impactor rails and specimen restraint. The bottom of the Boeing test fixture was covered by flexible foam rubber for insulation. The chamber was constructed with a removable cover to allow placement of the panels in the impact restraint fixture. A hole just large enough to allow the impactor to completely impact the panel was cut into the chamber cover. This hole was covered by a removable door to allow impact testing without having to remove the entire cover and start heating the panel. The temperature in the chamber was regulated by a temperature controller that cycled a cryogenic, normally closed valve, to release liquid nitrogen (LN_2) into the coil beneath the specimen. The end of the coil was plugged and holes were drilled in it along its length to release a fine spray of LN_2 beneath the restraint fixture. The LN_2 spray provided the necessary cooling for the subambient temperature testing. The temperature in the chamber was controlled to +/- 2 degrees F, which was verified by a separate thermocouple. Prior to impacting, the panel temperature was allowed to stabilize at -67 F by allowing a period of eight minutes of thermal soaking while

constrained in the test fixture. This was verified in a separate experiment, in which a thermocouple was placed within a piece of specimen material. This indicated that the interior temperature stabilized at -67 F within three minutes, which was well within the eight minute thermal soak period allowed prior to impact.

After the panel was placed in the impact restraint, the cover for the chamber was replaced and the panels were subjected to a single impact energy of 1500 in-lbs/in at -67 F. Repeated impacts, due to rebounding, were avoided by the use of a trigger and ratchet assembly to catch the impactor when it rebounded.
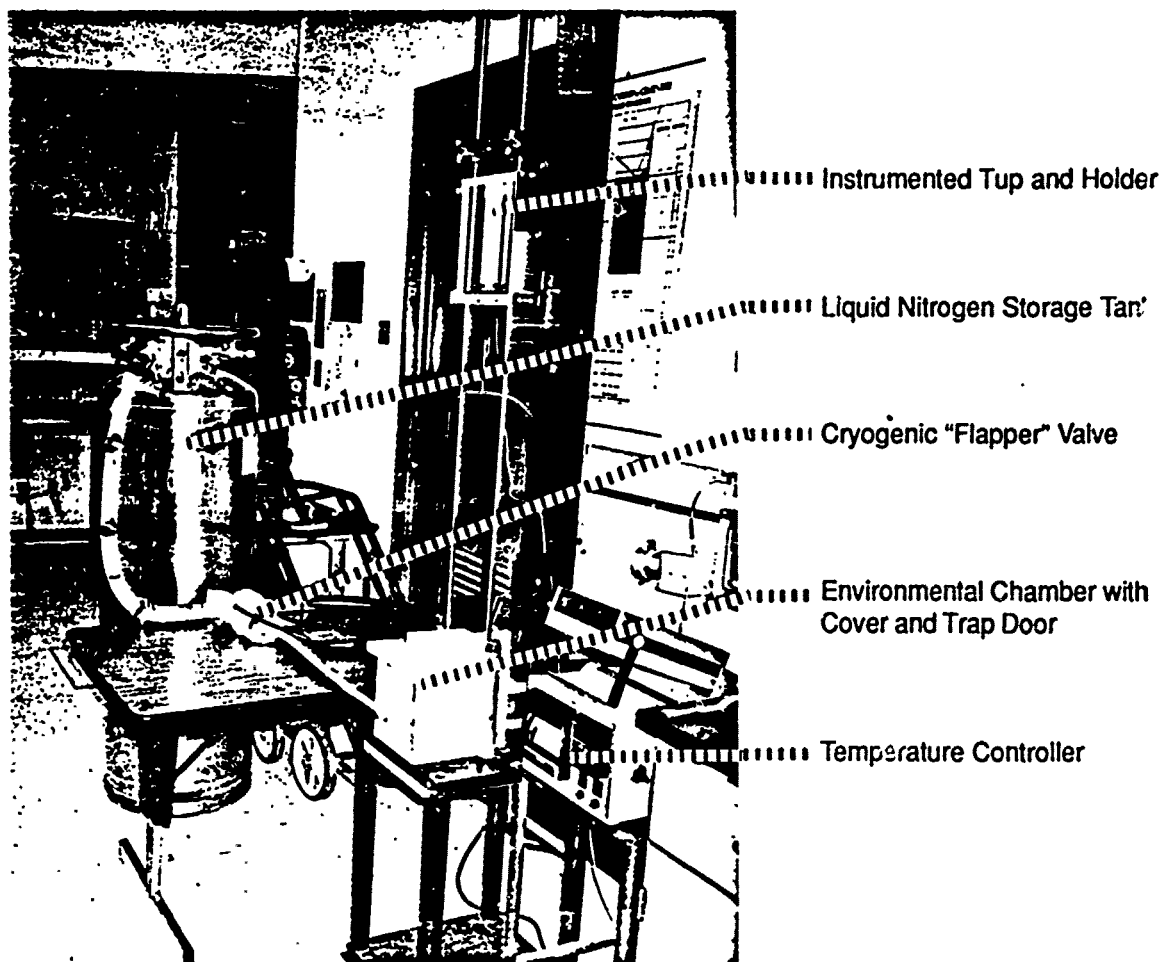
Instrumented Tup and Holder

Liquid Nitrogen Storage Tank

Cryogenic "Flapper" Valve

Environmental Chamber with
Cover and Trap Door
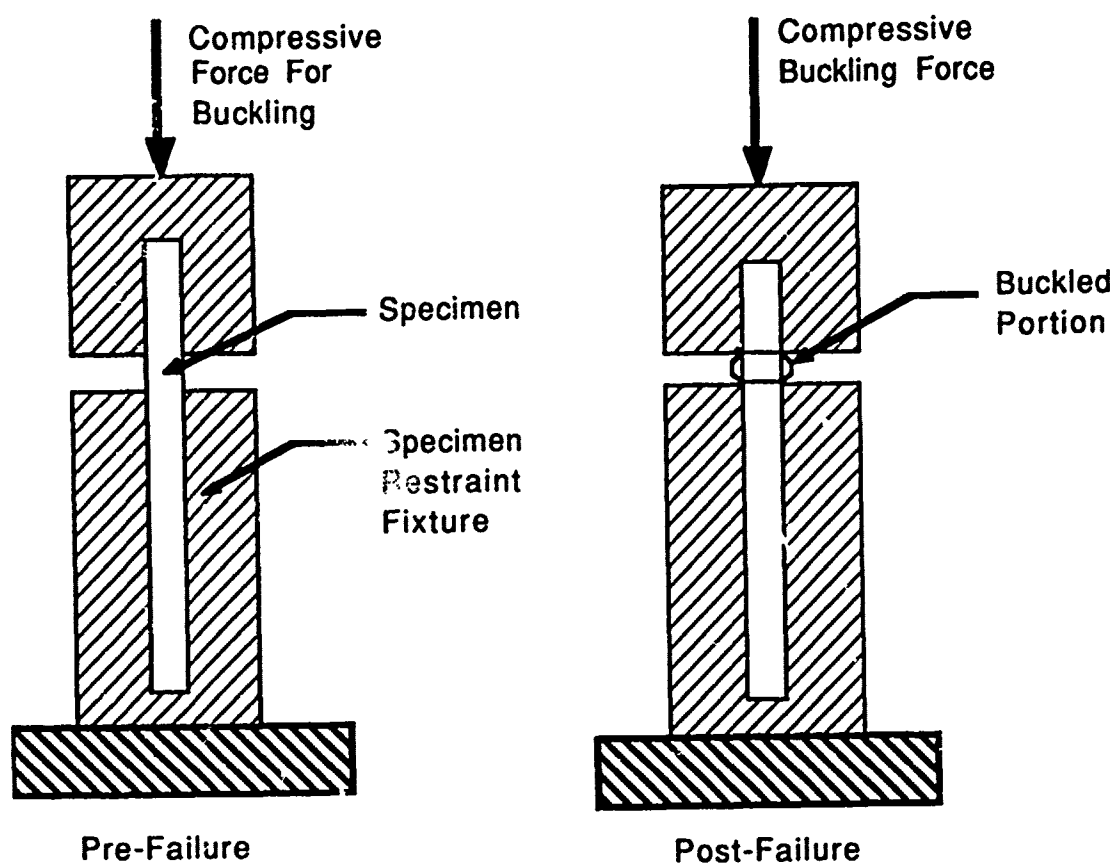
Temperature Controller

Figure 15 - Environmental Chamber

Post Impact C-scan

After the panels were impacted, they were allowed to warm to room temperature before being subjected to a post impact C-scan. The C-scan parameters were the same as those used for the preimpact ultrasonic scan in order to compare the difference between the pre and post impact C-scans.
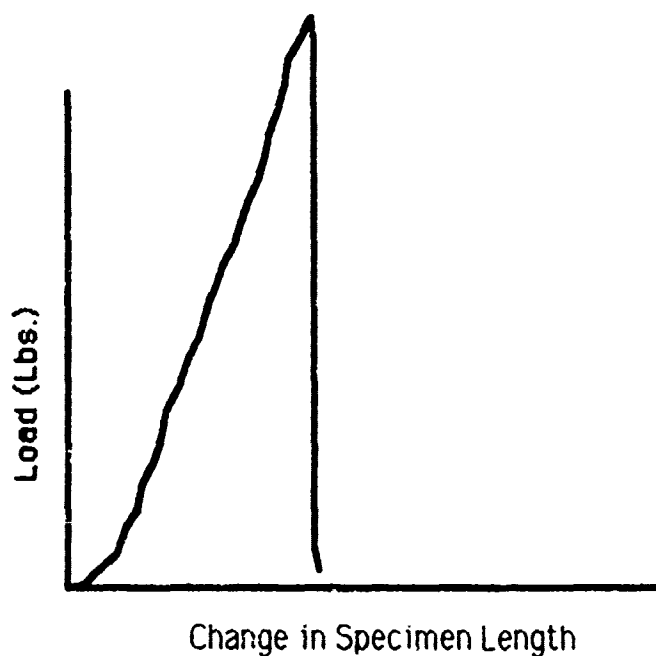
Compression After Impact Testing

Due to the unavailability of a compression restraint fixture, four of the six impacted specimens from each of the five plates were sent to Hercules Aerospace in Magna, Utah, where the CAI test was performed. This test is conducted because composite materials on the upper wing surfaces of aircraft, prone to damage from accidental tool drops, are subjected to uniaxial compression during flight. The test utilizes a fixture which constrains the edges of the specimen, except along a short length as shown in Figure 16. The panel is placed in uniaxial compression, along the zero degree ply axis, until buckling is detected by the strain gages in the unrestrained region of the panel. This buckling load is divided by the cross sectional area of the specimen to obtain the compressive strength. The results of this test are expressed in the pounds per square inch of cross sectional area required to cause buckling. A typical plot of the load versus deflection output is shown in Figure 17 below.

Compressive
Force For
Buckling

Compressive
Buckling Force

Specimen

Buckled
Portion

Specimen
Restraint
Fixture

Pre-Failure

Post-Failure

Figure 16 - Compression After Impact Test

Load (Lbs.)

Change in Specimen Length

Figure 17 - Example Plot From Compression After Impact Test

Fractography

In order to determine fracture details, such as mode and path, fractography specimens had to be prepared. One of the two remaining post impact specimens from each plate was slowly cut across the impact area with a diamond wafering blade, and potted in a clear epoxy potting resin. The potting resin was made of D.E.R. 332 epoxy resin from Dow Chemical Corp, and D230 Jeffamine and 399 Accelerator from Texaco. These were mixed in a weight ratio of 20 : 5 : 1, respectively. The potting compound allowed faster subsequent cutting rates without introducing cutting damage into the specimen. The potted specimen was cut into 0.075 inch thick slices in order to visually inspect how the damage propagated through the specimen, and fracture mechanisms present in different portions of the panel. Each slice was progressively polished by using 320, 400 , and 600 grit wet polishing cloth. This was followed by three minutes on an Polimet polishing wheel with 5 micron alumina polishing paste. The slices were inspected using reflected light microscopy. The different fracture mechanisms present in the panel will be shown in the results section.

Results and Discussion

Before discussion of the impact results, it is noted that the design and performance for the environmental chamber were more than adequate. The use of a proportional integral and derivative (PID) controller allowed the temperature to be rapidly lowered into the test range of -67 F. Once the chamber reached this subambient temperature, the PID controller cycled the LN_2 cryogenic valve to maintain this temperature +/- 2 deg F.

The average results from the instrumented impact testing of six specimens from each of the five plates are summarized in Figure 18. These show that the initial impact energy varied slightly to maintain the desired impact energy of 1500 in.-lbs/in. thickness. The increased specimen thickness of the modified panels, due to the two plies of FM300, required greater initial impact energies to attain 1500 in.-lbs/in. impact energy. Notice in the energy-time curve, Figure 13, the areas listed as the energy returned to the impactor, due to rebounding, and the energy absorbed in the specimen as damage. As previously stated, this can be compared to dropping a weight on a spring and compressing it beyond its elastic limit. The weight will not rebound off the spring with its same initial kinetic energy. The permanent deformation in the spring is analogous to the energy absorbed as damage in the composite panel. The normalized energy values listed in Figure 19 are essentially identical. This

indicates that this test did not detect any improvement, or degradation, in impact toughness due to the addition of the interleafs. The introduction of the interleafs was to minimize the amount of damage retained in the specimen as permanent damage. The above results indicate that this was not achieved.

	Thickness (in.)	Modification Location	Max Load (lbs)	Impact Energy (Ft-lbs)	Absorbed Energy (Ft-lbs)
Plate 1	0.1767	None	1664	22.73	19.21
Plate 2	0.1880	1st 0 Degree	1688	23.98	20.71
Plate 3	0.1899	2nd 0 Degree	1737	24.18	20.08
Plate 4	0.1884	3rd 0 Degree	1683	24.01	20.07
Plate 5	0.1882	4th 0 Degree	1696	24.05	20.31

Figure 18 - Average Impact Test Results
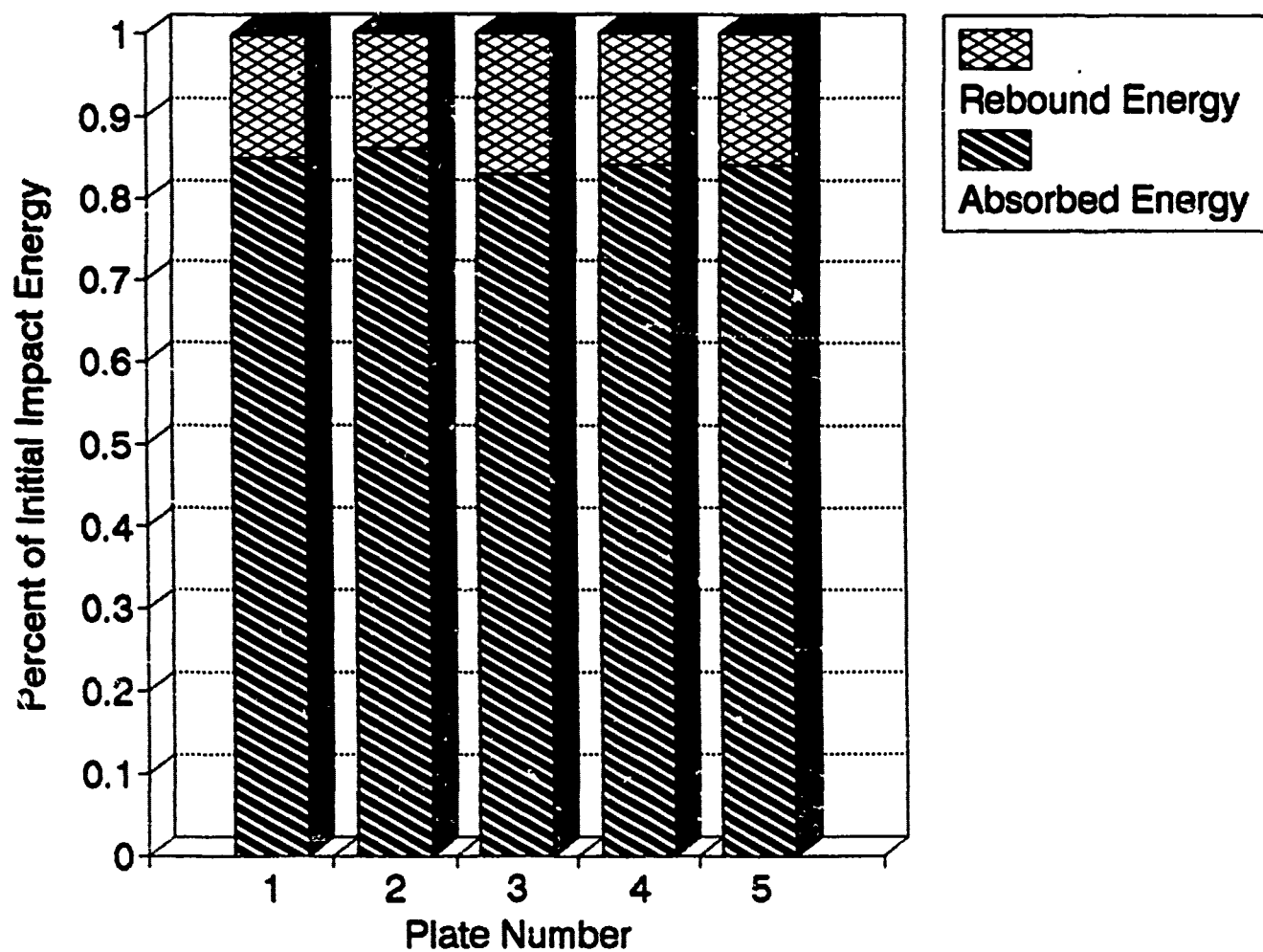
Figure 19 - Instrumented Impact Absorbed Energy Results

The comparison of the post impact C-scan inspections of the panels, shown in Figure 20, indicates a decrease in damage area in the interleafed panels. The increased impact toughness of the FM300 interleafs helped absorb a portion of the impact energy, thus preventing damage. The decreased damage area in the modified plates is important for two reasons. As the technology becomes available to completely repair the impact damaged areas, a smaller damage area would be less expensive to repair and require less downtime for the aircraft. The second reason is the decreased damage area would be less susceptible to fatigue failure before it was repaired. Although the effects vary from panel to panel, there appears to be a slight advantage when the interleaf is located over the first zero degree ply for the greatest reduction in impact damage area.
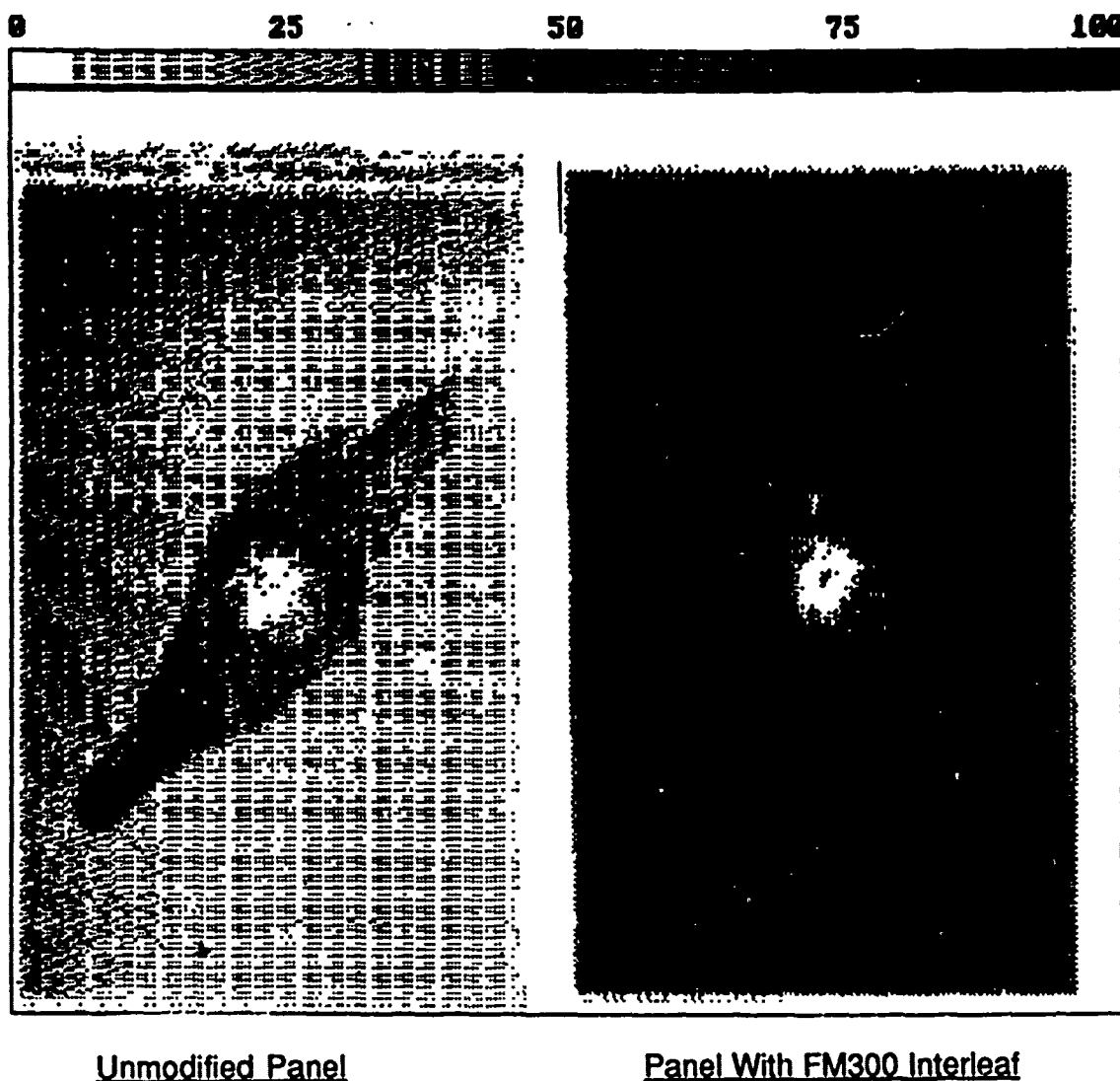
Figure 20 - C-scan Comparison of Damage Area.

The compression after impact test results, Figure 21, show that the failure loads of the unmodified panels were within the normal range for IM6/3501-6 materials, as indicated by the room temperature data

for the unmodified configuration, listed as the control panel. The average thickness of 0.1767 inch is also within the normal range for panels constructed of 145 gm/m^2 prepreg material. An interesting note is that the unmodified IM6/3501-6 system had the same post impact compressive strength regardless of impact temperature. This suggests that the unmodified IM6/3501-6 system appears to be relatively unaffected by low temperature embrittlement. The average CAI strengths of the interleafed panels were slightly lower than the unmodified panel. The average buckling loads for the each of the panels were relatively constant, so when the additional thickness of the interleafs was subtracted, the CAI strength of the modified panels were identical to the unmodified panels. No optimum interleaf location was found that maximized the post impact compressive strength since the interleafs actually tended to slight. ' decrease the post impact compressive strength of the panels.
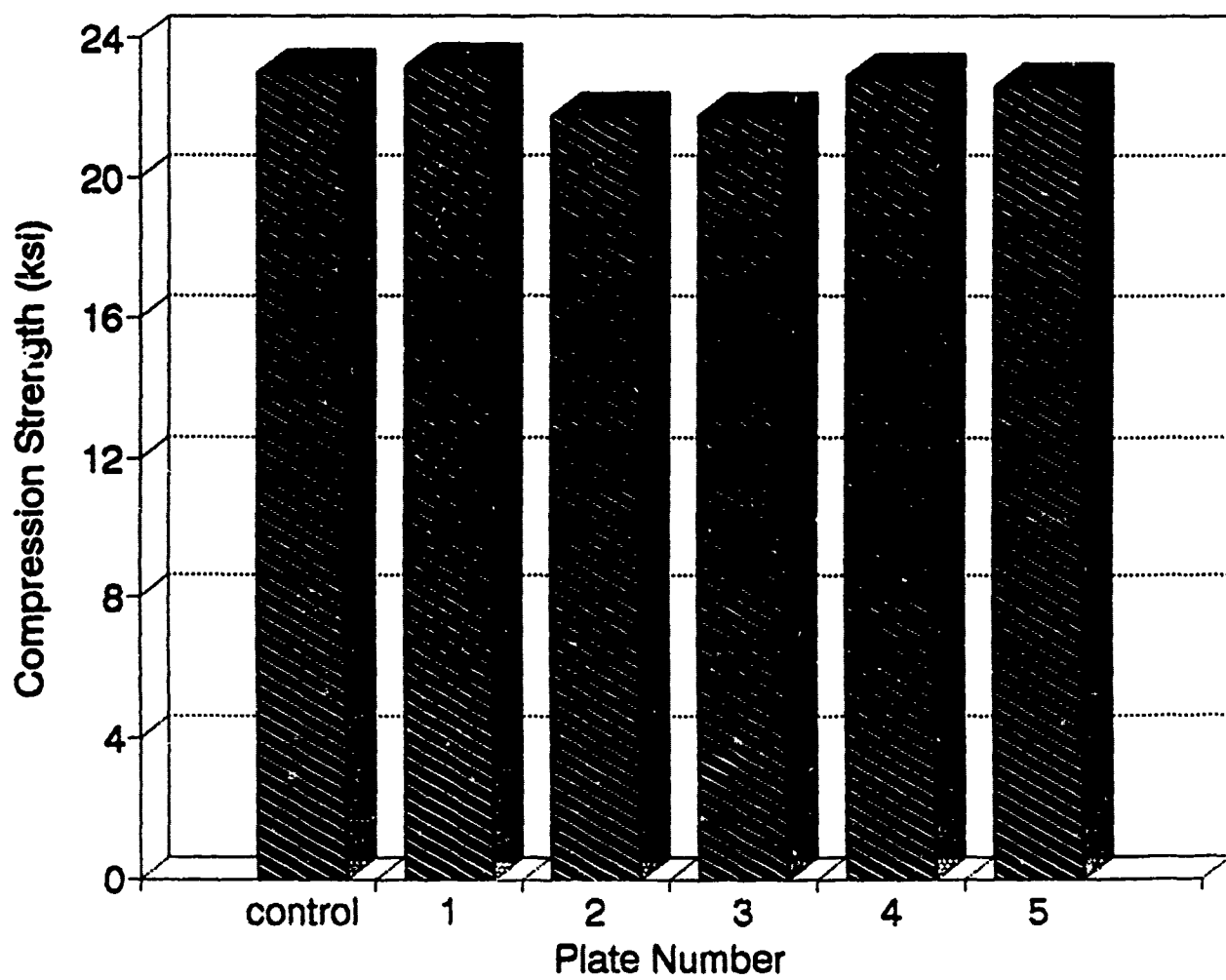
Figure 21 - Compression After Impact Test Results

Representative fracture damage found in the impact damaged specimens is shown in Figure 22. The fractographic analysis indicated a large amount of delamination, debonding, fiber breakage, and matrix cracking in the impact damaged area. There was no clear localized pattern of delamination or debonding near the interleaf, which would indicate poor bonding between the interleaf and the epoxy matrix. The impact energy caused a large amount of fiber breakage and matrix cracking as shown in Figure 22.
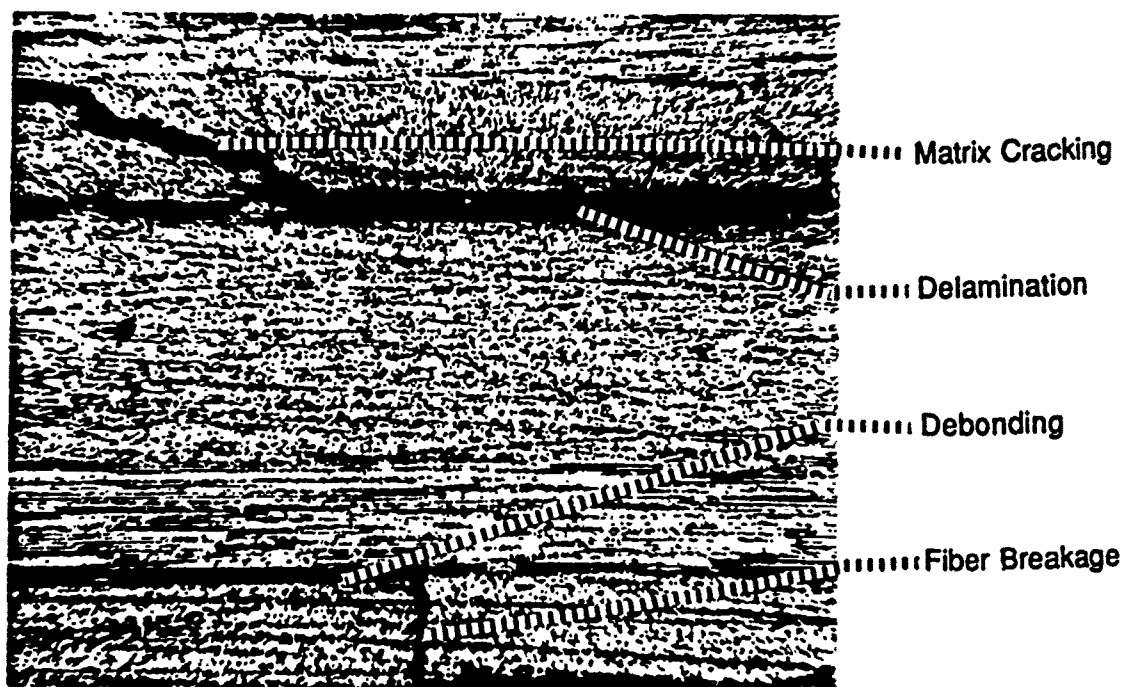
Post Impact Specimen @ 100X

Figure 22 - Representative Damage Types in Post Impact Specimen

Conclusions

The following important observations are made.

(1) The unmodified IM6/3501-6 system appears to be unaffected by low temperature embrittlement.

(2) At -67 deg F, the interleafed FM300 plies in each panel slightly decreased the post impact compressive strength of the material compared to the unmodified control panel.

(3) There was no optimum interleaf location found that increased the post impact compressive strength.

(4) The interleafs did decrease the damage area compared to the unmodified panels, as characterized by C-scan inspections

(5) There was an optimum location for the interleafs that yielded the greatest reduction in impact damage area.

(6) The low temperature impact of unmodified IM6/3501-6 did not produce the predicted decrease in post impact compressive strength when compared to room temperature impact of identical specimens.

(7) Although studies have shown that the interleafing of tough components has increased the post impact compressive strength at room temperature, the use of FM300 interleafs in low temperature environments does not increase the post impact compressive strength.

(8) Fractographic analysis of the subambient temperature impact specimens does not indicate any clear reasons why the FM300 interleafs do not reduce the impact damage to a greater extent.

References

1.   Schwartz, M.M., *Composite Materials Handbook*, McGraw-Hill, New York, 1983, p 7.22.

2.   Exodus 5:7, *Today's English Version*, American Bible Society, New York, New York, 1986.

3.   Ashton, J.E., Halpin, J.C., and Petit, P.H., *Primer on Composite Materials: Analysis.*, Technomic Publishing Co, Inc., Stamford, Conn, p.2.

4.   Schwartz, M.M., p 1.2.

5.   Boll, D. J., and Bascom, J. C., "A Microscopy Study of Impact Damage of Epoxy-Matrix Carbon-Fiber Composites," *Journal of Materials Science*, Vol. 21, 1986, p. 2667.

6.   Williams, J. G., and Rhodes, M. D., "The Effect of Resin on the Impact Damage Tolerance of Graphite-Epoxy Laminates," *NASA TM 83213*, Langley Research Center, Hampton VA 23665, Oct. 1981.

7.   Byers, B. A., "Behavior of Damaged Graphite/Epoxy Laminates Under Compression Loading," *NASA-CR-159293*, National Aeronautics and Space Administration, Jan. 1980.

8.   Griffin, C.F., "Damage Tolerance of Toughened Resin Graphite Composites," *Toughened Composites,* ASTM STP 937, Norman J. Johnston, Ed., American Society for Testing and Materials, Philadelphia, 1987, p. 23.

9.   Jordan, W.M. and Bradley, W.L., "Micromechanisms of Fracture in Toughened Graphite-Epoxy Laminates," *Toughened Composites,* ASTM STP 937, Norman J. Johnston, Ed., American Society for Testing and Materials, Philadelphia, 1987, p. 96.

10.   Sohi, M. M., Hahn, H. T., and Williams, J. G., "The Effect of Resin Toughness and Modulus on Compressive Failure Modes of Quasi-Isotropic Graphite/Epoxy Laminates," *Toughened Composites,* ASTM STP 937, Norman J. Johnston, Ed., American Society for Testing and Materials, Philadelphia, 1987, p. 38.

11. Kam, C. Y. and Walker, J. V., "Toughened Composites Selection Criteria," *Toughened Composites,* ASTM STP 937, Norman J. Johnston, Ed., American Society for Testing and Materials, Philadelphia, 1987, p. 12.

12. Hirschbuehler, K. R., "A Comparison of Several Mechanical Tests Used to Evaluate the Toughness of Composites," *Toughened Composites,* ASTM STP 937, Norman J. Johnston, Ed., American Society for Testing and Materials, Philadelphia, 1987, p. 73.

13. Evans, R. E. and Masters, J. E., "A New Generation of Epoxy Composites for Primary Structural Applications: Materials and Mechanics," *Toughened Composites,* ASTM STP 937, Norman J. Johnston, Ed., American Society for Testing and Materials, Philadelphia, 1987. p. 428.

14. American Cyanamid Co., *Structural Adhesives Selector Guide,* American Cyanamid Co., Havre de Grace, MD, 1989, p. 12.

15. Hercules Incorporated, *Product Data Sheet Number 865,* Hercules Incorporated, Wilmington, DE.

16. Krautkramer, Josef, *Ultrasonic Testing of Materials,* 2nd. Ed, Springer-Verlag Berlin, 1977, p. 26.

17. Filipczynski, Leszek, Pawlowski, Zdzislaw, and Wehr, Jerzy, *Ultrasonic Methods of Testing Materials,* Butterworths, London, 1966, P. 6.

18. General Research Corp., *Dynatup GRC 730-I Instruction Manual,* General Research Corp, Goleta, CA, p. 94.

19. Visconti, I Crivelli, "The Study of the Behavior of Composite Materials Using Ceast Instrumented Impact Testing", Symposium on Fracture Mechanics, Bradford University, April 20, 1983, p. 4.

20. Suppliers of Advanced Composite Materials Association, *SACMA recommended Test Method SRM 2-88.for Compression After Impact Properties of Oriented Fiber - Resin Composites,* SACMA, 1988.